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RESEARCH MEMORANDUM

PROBLEMS OF PERFORMANCE AND HEATING OF HYPERSONIC VEHICLES

By H. Julian Allen and Stanford E. Neice

Ames Aeronautical Laboratory
Moffett Field, Calif.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

March 5, 1956

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RESEARCH MEMORANDUM

PROBLEMS OF PERFORMANCE AND HEATING OF HYPERSONIC VEHICLES

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INTRODUCTION

A particular virtue of high-velocity rockets for military application is the difficulty in effecting countermeasures for defense against them. In consequence, Sänger (ref. 1) devoted considerable effort to the study of several types of rocketcraft and, with Bredt (ref. 2), examined the rocket-powered glider airplane in particular.

There are two principal objections to the rocketcraft when compared to the conventional supersonic airplane powered by air-breathing engines. First, the propulsive efficiency of the chemical rocket is low so that the "all-up" weight at take-off is large in comparison with that at rocket burnout. Second, since the long-range rockets attain very high speeds, they may accordingly be subjected to intense aerodynamic heating when in the atmosphere. Often the heating rates are so great as to preclude the possibility of adequate cooling by radiation alone, in which case they must be protected by providing a sufficient weight of coolant to absorb this heat. At best, the weight of coolant required may be of the order of the payload weight which only serves to amplify the importance of the first objection since, then, a large increase in the initial all-up weight must be provided simply to propel the coolant. In fact, it is possible in certain cases to generate so much heat that no payload can be carried at all since all the available weight at rocket burnout is required for coolant to cool the coolant. Thus, it is seen, as with so many problems in aircraft design, that the cooling problem has a pyramiding nature and is consequently of extreme importance.

From what has been said, it is clear that there are two closely connected questions which the designer must ask himself: "Can the rocket vehicle be made reasonably efficient compared with the airplane?" and "What can be done to minimize the aerodynamic heating problem?"

In this paper, three types of hypervelocity vehicles are compared: the ballistic rocket, the skip rocket, and the rocket glider, in a manner somewhat similar to that originally done by Sänger. The trajectories of these rockets are shown in figure 1. The ballistic vehicle considered here is the one which leaves the atmosphere at that angle relative to the earth's surface which requires the least energy input for a given flight

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range. The skip vehicle travels on a succession of ballistic trajectories, each connected to the next by a "skipping phase" during which the vehicle enters the atmosphere, negotiates a turn, and then is ejected from the atmosphere. The optimum skip vehicle is considered in which the skipping phase of flight is made at the optimum lift-drag ratio and the initial flight angles of the successive ballistic trajectories are those which, for the optimum lift-drag ratio, yield the given flight range for the least energy input. The boost-glide vehicle considered is one which during the powered and the unpowered phase of flight flies in the atmosphere at optimum lift-drag ratio for each point of the trajectory. Thus the flight altitude continuously varies with speed.

These three hypervelocity rockets will be compared efficiency-wise with one another and with assumed supersonic jet airplanes. Next, the hypersonic vehicles will be compared on the basis of aerodynamic heating requirements. Finally, some detailed problems of glide rockets will be discussed.

RANGE EFFICIENCY

In order to compare the efficiency of flight of the various vehicles, it should be apparent at the outset that the efficiency parameter which is truly appropriate depends upon the intended use of the vehicles. Thus, for a missile, the parameter of real interest is the vehicle cost per pound of explosive delivered; whereas, for the usual transport, which is not destroyed on completion of a single mission, the proper parameter would be the total cost of fuel, repairs, and depreciation per pound of payload delivered.

In both cases, these parameters might well be approximated by the ratio of initial weight at take-off to the payload weight. However, the evaluation of this ratio requires a knowledge of the weight of the component parts of the structures which is a matter of detail design beyond the scope of this paper. Accordingly, in this paper the ratio of the initial weight at take-off to the final weight after fuel is expended will be used as the measure of flight efficiency. It is presumed, then, that the reader will temper the results given in the following discussion with the knowledge that the ratio of payload weight to final weight is not the same for the several classes of vehicles considered. In particular, it should be noted that the use of the ratio of initial weight to final weight as the measure of merit is particularly unfair to the ballistic vehicle since its ratio of payload weight to final weight is generally much greater than for the other types.

In order to compare the efficiency of the hypersonic rockets and the airplane, it is desirable to express the range equation in a form of the type first developed by Bréguet, because in this form it is most familiar to the airplane designer. In order to effect such a form for the range equation, the following mathematical development is employed. (See ref. 3.) The effective drag D_e is defined in such a manner that the product of this drag and the range X equals the energy input at burnout as follows:

$$D_e X = \frac{W_f V_f^2}{2g} \tag{1}$$

where V_f is the final speed at rocket burnout, W_f is the corresponding weight, and g is the acceleration due to gravity. Let the effective lift L_e be defined as the weight at rocket burnout; that is,

$$L_e = W_f \tag{2}$$

Combining these two equations then gives the range as

$$X = \left(\frac{L}{D}\right)_e \frac{V_f^2}{2g} \tag{3}$$

The term $(L/D)_e$ will be called the effective lift-drag ratio herein.

Now, the speed at rocket burnout V_f may be related to the ratio of initial weight W_i to final weight W_f in the form

$$V_f = I_e g \, ln \frac{W_i}{W_f} \tag{4}$$

where I_e is defined as the effective specific impulse of the rocket propellant which is generally somewhat less than the actual specific impulse because of the requirements of staging and so forth.

If one of the V_f's in equation (3) is replaced by the value from equation (4), the range can be obtained in the form

$$X = \left(\frac{L}{D}\right)_e I_e V_e \ ln \frac{W_i}{W_f} \tag{5}$$

wherein the effective speed V_e is just one-half the speed at rocket burnout. For a conventional airplane with air-breathing engine, the Bréguet equation can be written in the form

$$X = \frac{L}{D} \, I \, V \ ln \frac{W_i}{W_f} \tag{6}$$

where L/D is the aerodynamic lift-drag ratio. It is more usual with airplanes to replace the product IV by the product of the thermal efficiency η and specific-heat value of the fuel h to give

$$X = \frac{L}{D} \, \eta h \ ln \frac{W_i}{W_f} \tag{7}$$

Equation (5) can now be used for comparing the hypervelocity rockets with one another, and these rockets in turn may be compared with the conventional airplane with the use of equation (6) or (7).

Obviously, the most efficient vehicle, based on the definition given, is the one with the largest value of the product $(L/D)(I)(V)$; and this product may be broken down for convenience into the components L/D, which is the measure of aerodynamic efficiency, and IV, which is the measure of propulsive efficiency. In figure 2 is shown a comparison of conventional air-breathing-engine propulsive systems with a typical chemical-rocket system as indicated by the attainable values of I V as a function of speed. Since this product has the dimensions of a length, it has been termed the "propulsive range." This range has been expressed in nautical miles. It is seen that, depending on the speed chosen, the air-breathing-engine system will attain an optimum value of I V which is essentially independent of the speed as has been pointed out by Rutowski (ref. 4) and others. This value is about 700 nautical miles. For rockets, the specific impulse is essentially a constant (assumed herein as 225 seconds) and hence the product has the linear characteristics shown. The high value of 700 nautical miles would not be approached until burnout speed corresponding to the escape speed from the earth is reached. Thus, the rocket has the disadvantage that its propulsive efficiency for normal ranges is low. This is not the whole story, however, since it is the product of propulsive range and the effective lift-drag ratio which is important.

In order to examine the effective lift-drag ratios for the rockets and for the airplane, figure 3 has been prepared. In this figure, values are shown as a function of range in nautical miles. Figure 3(a) is for the case wherein the aerodynamic lift-drag ratio is 2 and figure 3(b) is for the case wherein the ratio is 6. For the airplane, the lift-drag ratio is independent of range, but, for the rockets, it is seen that the effective lift-drag ratio increases with increasing range. This anomalous result occurs because the increased range for the rockets is obtained through increased speed so that an increasingly greater share of the vehicles' weight is borne by centrifugal force on the curved flight around the earth. Thus the aerodynamic lift required decreases and hence also the aerodynamic drag. In fact, when satellite speed is reached, the effective lift-drag ratio becomes infinity. It is of particular interest to note that the ballistic vehicle for which the actual aerodynamic lift-drag ratio is zero behaves practically as though it were a rocket glider having an aerodynamic lift-drag ratio of 2.

If the effective lift-drag ratio is now combined with the propulsive range, the results shown in figure 4 are obtained. The ratio of initial weight to final weight as a function of range for the four types of vehicles is shown in figure 4(a) for an aerodynamic L/D of 2 and in figure 4(b) for an aerodynamic L/D of 6. Here it is noted that all the hypervelocity vehicles look attractive when the flight range is long and the attainable aerodynamic lift-drag ratios are low. Notice also that if the aerodynamic L/D attainable is small, the skip rocket appears to be the best of the hypervelocity vehicles; but, if the aerodynamic L/D is large, then the skip and glide rockets are about equal. It should also be pointed out that, although the ballistic rocket looks poor in figure 4(b), in general, compared to the others, as noted earlier, a larger fraction of its final weight is payload because of the low engine and fuel tankage weight. Hence, if the ratio of initial weight to payload weight had been used as a measure of merit, the ballistic vehicle would appear promising.

AERODYNAMIC HEATING

It was pointed out earlier that all hypervelocity vehicles are subjected to intense aerodynamic heating. If the aerodynamic heat must be absorbed in a coolant and the required coolant weight becomes large, then the all-up weight at take-off can become very great compared with the payload weight. Thus the two questions which must be asked about each of the three types of rockets are: (1) "Is it possible to radiate the incoming aerodynamic heat at a sufficiently high rate to keep the temperature within allowable bounds?" and (2) "If not, can the required quantity of heat which must be absorbed be kept small enough to prevent the necessity of an excessive weight of coolant?"

In order to answer the first question, figure 5 has been prepared to show the maximum value of the average heat transfer rate for each of the three rockets as the speed, and hence the range, is increased. The transfer of kinetic energy to heat occurs in abrupt pulses during the skip phases for the skip rocket. The first-skip heat rate, which is the most severe, is shown here. The ballistic rocket experiences the heat in a single abrupt pulse on atmospheric entry. The glide rocket, on the other hand, gradually converts its kinetic energy to heat over the whole flight trajectory. Thus, the relatively low rate of heat input is not surprising. Also shown in this figure is the rate of heat input for radiation equilibrium at temperatures of 1,000° F, 2,000° F, and 3,000° F. The answer to the first question is clear. The ballistic and skip rockets that are being considered cannot possibly be satisfactorily cooled by thermal radiation alone except for short flight ranges and hence must rely on a coolant. The radiation method of cooling does seem feasible for the glide rocket, although barely so.

The second question is now - "For the ballistic and skip rockets, which appear incapable of being cooled by radiation, can the total heat input be kept sufficiently low so that excessive weight of coolant is not required?" In order to answer this question, first consider the case of the ballistic rocket. In figure 6 is shown the total heat input for a 5,000-pound conical ballistic warhead as a function of cone angle. The chosen base area is 10 square feet and the velocity at atmospheric entry is 20,000 feet per second. It is seen that for turbulent flow there is a pronounced reduction of the heat input with increase in cone angle. The heat input is low for all but the smallest cone angles for the laminar case. The reason for the pronounced reduction of heat input with cone angle for the turbulent flow case is the following (ref. 5): For the warhead weights of usual interest, the kinetic energy near impact is a small fraction of the kinetic energy that the vehicle had on entering the atmosphere. Hence, nearly all this energy must be converted to heat but the fraction of this heat which enters the warhead is proportional to the ratio of the friction drag to the total drag. The remainder of the energy is spent heating the atmosphere. Thus, by making the ratio of friction drag to total drag small - in this case by employing large cone angles - the total heat input is kept small. The question naturally then follows: "Why doesn't a similar large reduction of heat input with cone angle occur in the laminar flow case?" The answer can be gotten from figure 7 wherein the maximum Reynolds number, which is also a measure of the mean Reynolds number, is plotted as a function of cone angle for the cones considered in figure 6. It is seen that there is a large reduction in Reynolds number with cone angle. This change in Reynolds number does not have a very pronounced effect on the turbulent friction coefficient, since this friction coefficient is only a weak function of the Reynolds number. For laminar flow, on the other hand, the friction coefficient varies inversely as the square root of the Reynolds number. Thus the friction coefficient drops rapidly with decrease in cone angle and hence the ratio of friction drag to total drag tends to stay more nearly uniform with cone angle, which explains the behavior of the heat input with cone angle for the laminar case shown in figure 6. Referring again to figure 7, if past experience at lower speeds is typical of the state of affairs at high speeds, it is most unlikely that laminar flow can be maintained at the very high Reynolds numbers associated with the entry of the small angle cones. It is very doubtful that even for the large angle cones continuous laminar flow will occur, but it is probable that during the initial portion of the entry trajectory, when the Reynolds numbers are much less than the maximums shown in figure 7, long runs of laminar flow can be maintained. It is during this initial flight trajectory that the laminar flow is particularly desired since then the flight speed is greatest so that time rates of heat input tend to be most severe. At best, then, the "high drag" solution to the heating problem for the ballistic vehicle would seem to be the most logical course to follow. However, it should be expected that the total heat input will be something between that for all-laminar and all-turbulent flow.

Unfortunately, this "high drag" solution is not open to the skip rocket. This conclusion follows directly from the fact that the skip rocket must develop reasonably high lift-drag ratios to achieve long range. But inasmuch as it is known that high lift-drag ratios are incompatible with high pressure drag, the skip rocket will clearly be relatively slender and consequently will have a relatively high ratio of friction drag to total drag.

Now, the question as to whether the heat input to the ballistic and skip rockets can be kept sufficiently low can be answered. In figure 8 is shown the calculated convective heat input per unit weight for a conical ballistic rocket having a large cone angle (60°) and the convective heat input per unit weight during the first skip for a conical skip rocket having a sufficiently small cone angle to permit a lift-drag ratio of 6. The flow in both cases is assumed laminar. In spite of the fact that the total energy converted to heat in the first skip of the skip rocket is much less than that involved in the entry of the ballistic rocket, the ratio of friction drag to total drag for the skip rocket is so large relative to that for this ballistic rocket that the heat input is seen to be much greater. Thus the ballistic vehicle could be cooled with a not-too-great weight of copper (see dashed curve) as a coolant, but it is doubtful that this skip rocket could be satisfactorily cooled at all, except for very short range flight. Thus when heating is considered, only the glide rocket (which can, in the main, be cooled by radiation) and the ballistic rocket (which is not required to accept an inordinate amount of heat) appear attractive hypersonic types at this time.

What has been said about aerodynamic heating up to this point applies to an average surface element of these vehicles. Of perhaps even greater importance is the heating of particular local surface elements where the heat rates may be many times that for the average surface. Such local elements are commonly the stagnation points of bodies and the leading edges of wings. It should be apparent that pointed noses or sharp leading edges seem impractical as regards aerodynamic heating since not only is the capacity for heat retention small but the heat transfer rate is exceedingly high since it varies nearly inversely as the square root of the radius of curvature. Thus a truly pointed nose or sharp leading edge would ablate, melt, or burn away.

For the ballistic warhead no problem arises in blunting the nose since the important effect of the blunting that may be required is to increase the pressure drag which is a desirable feature as has previously been discussed. (This does not consider possible adverse effects that excessive blunting may have on the transition from laminar to turbulent flow.)

PROBLEMS OF GLIDE ROCKETS

For the glide vehicle, the highest possible lift-drag ratio is urgently desired so that the drag incurred by blunting must be kept to a minimum. For the fuselage nose, slight blunting has been found not to increase the drag, but, for the wing, even a slight blunting is deleterious. However, theoretical and experimental research has shown that the drag increment can be kept low by use of swept leading edges. In fact, it can be shown (see ref. 6) that for a given rate of heat input the drag due to blunting of the leading edge varies approximately as the fourth power of the cosine of the sweep angle. For this and other reasons, one suggested configuration for a man-carrying boost-glide rocket might well look like the configuration shown in figure 9 (ref. 7). In the case of a man-carrying glider, a certain minimum span will be required for landing. The maximum leading-edge sweep will thus be obtained if the leading edge runs from the fuselage nose to the end of the span opposite the fuselage base. For the case shown in figure 9, the leading-edge sweep is 74° and it has been calculated that, at Mach numbers up to the order of 7, the drag due to the required blunting of the leading edge is not large and, for a 50-foot-long vehicle, the lift-drag ratio should be 5 if the boundary-layer flow is turbulent and should be 6 if the boundary-layer flow is laminar. Laminar flow is doubly desirable since it both improves flight efficiency and reduces aerodynamic heating. Now, it should be noted that, for speeds not too near orbital speed, flight at constant lift-drag ratio infers nearly constant dynamic pressure; hence, as shown in figure 10, the Reynolds number decreases with increase in speed. It becomes zero at orbital speed since centrifugal force is all that is needed to support the weight. On the other hand, recent experimental research has shown that the transition Reynolds number generally increases with increasing speed. In fact, in the Ames supersonic free-flight tunnel, continuously laminar flows have been maintained at a Mach number of 7 on bodies of revolution with relatively rough surfaces to Reynolds numbers of the order of 15×10^6 - which is of the order shown here. Thus, it is not surprising that in some recent firings of a model of the three-wing configuration at essentially full-scale Reynolds numbers, the indicated lift-drag ratio was 5.5. Although it is true that this rather high lift-drag ratio can be attained up to Mach numbers of the order of 7, a boost-glide vehicle having this maximum Mach number will have a range of only about 800 nautical miles. In order to increase the range, the Mach number must be increased, but, in so doing, the required leading-edge bluntness must be increased to prevent excessive heating. It can readily be found that the drag incurred by the blunting can then become so large as to reduce the lift-drag ratio seriously. It was shown earlier that the product of $\frac{L}{D} I V$ was a measure of the flight efficiency. Thus, for a given value of L/D, the efficiency improves with increasing speed. On the other hand, if the lift-drag ratio decreases with increasing speed, it is possible for the efficiency to diminish as the range is increased. In this event, rather than to employ the simple boost-glide trajectory for the rocket airplane in which the maximum boost is maintained to give the full speed required for the desired range, it would be preferable to boost to a somewhat lower speed and then sustain this speed with a lower thrust rocket for the distance required to obtain this same range. In this case, the reduced leading-edge radius might well improve the lift-drag ratio enough to more than make up for the reduction in the propulsion efficiency. This situation in which the leading-edge stagnation temperature is restricted to $2,000^\circ$ F is indicated in figure 11 where range is plotted as a function of the ratio of initial weight to final weight for the three-wing glider shown previously. Each of the individual solid curves corresponds to a particular leading-edge radius. The circled end points correspond to simple boost-glide flight (that is, no sustainer) while the higher values of each curve correspond to increased amounts of sustainer flight for the increased range. The dashed envelope curve, which represents the optimum performance, shows that some sustainer portion of flight is desired when the leading-edge temperature is limited.

If one determines such envelope curves for various leading-edge temperatures, it is possible to express optimum weight ratio as a function of range for various permitted leading-edge temperatures as is shown in figure 12. It is seen that for the larger ranges there is a weight penalty when the radiation equilibrium temperature of the leading edge is limited. Whether some cooling by other means than radiation should be used would depend on how the weight penalty for coolant compares with the penalty shown in figure 12.

In conclusion, it is pertinent to examine what further can be done to improve flight efficiency for glide rockets. First, it is obvious that, since aerodynamic heating appears to preclude the use of the very high speeds required to obtain good efficiency with a simple rocket (see fig. 2), some real effort should be made to develop an engine such as the rocket-ramjet to improve the propulsive range for Mach numbers from 5 to 10. Second, every effort should be made to improve the aerodynamic lift-drag ratio. In this regard, tests have recently been made of configurations of the type shown in figure 13 in which the body bow wave has been used to assist in providing lifting pressures under the wing. The negative dihedral at the tips is not only used to provide directional stability but also to turn the outflow from the body downward to enhance the lift further and so improve the lift-drag ratio. The calculated variation of lift-drag ratio with Reynolds number, with laminar flow assumed, is given in figure 14 for this configuration with the design Mach number of 5. The experimental value of the lift-drag ratio, obtained at a Reynolds number based on body length of 2.5×10^6 in the Ames 10- by 14-inch tunnel, is, as shown, 6.35. This value agrees fairly well with the calculated value of 6.81. At flight Reynolds number, lift-drag ratios of the order of 10 should thus be obtainable. Even with such a high lift-drag ratio, it is important to note that the largest component of the drag is skin friction. It is clear, then, that research should be directed to find ways to reduce the magnitude of the friction drag. Perhaps, for example, the use of transpiration cooling through porous surfaces, which theory indicates (ref. 8) will result in a reduction of the average friction coefficient, should be considered.

Ames Aeronautical Laboratory
National Advisory Committee for Aeronautics
Moffett Field, Calif., Dec. 15, 1955

REFERENCES

1. Sänger, Eugen: Raketen-Flugtechnik. R. Aldenbourg (Berlin), 1953.

2. Sänger, Eugen, and Bredt, J.: A Rocket Drive for Long Range Bombers. Translation CGD-32, Tech. Information Branch, Bur. Aero., Navy Dept., Aug. 1944.

3. Eggers, Alfred J., Jr., Allen, H. Julian, and Neice, Stanford E.: A Comparative Analysis of the Performance of Long-Range Hypervelocity Vehicles. NACA RM A54L10, 1955.

4. Rutowski, Edward S.: Energy Approach to the General Aircraft Propulsion Problem. Jour. Aero. Sci., vol. 21, no. 3, Mar. 1954, pp. 187-195.

5. Allen, H. Julian, and Eggers, A. J., Jr.: A Study of the Motion and Aerodynamic Heating of Missiles Entering the Earth's Atmosphere at High Supersonic Speeds. NACA RM A53D28, 1953.

6. Eggers, A. J., Jr., Hansen, C. Frederick, and Cunningham, Bernard E.: Theoretical and Experimental Investigation of the Effect of Yaw on Heat Transfer to Circular Cylinders in Hypersonic Flow. NACA RM A55E02, 1955.

7. Seiff, Alvin, and Allen, H. Julian: Some Aspects of the Design of Hypersonic Boost-Glide Aircraft. NACA RM A55E26, 1955.

8. Rubesin, Morris W.: An Analytical Estimation of the Effect of Transpiration Cooling on the Heat-Transfer and Skin-Friction Characteristics of a Compressible, Turbulent Boundary Layer. NACA TN 3341, 1954.

HYPERSONIC VEHICLES

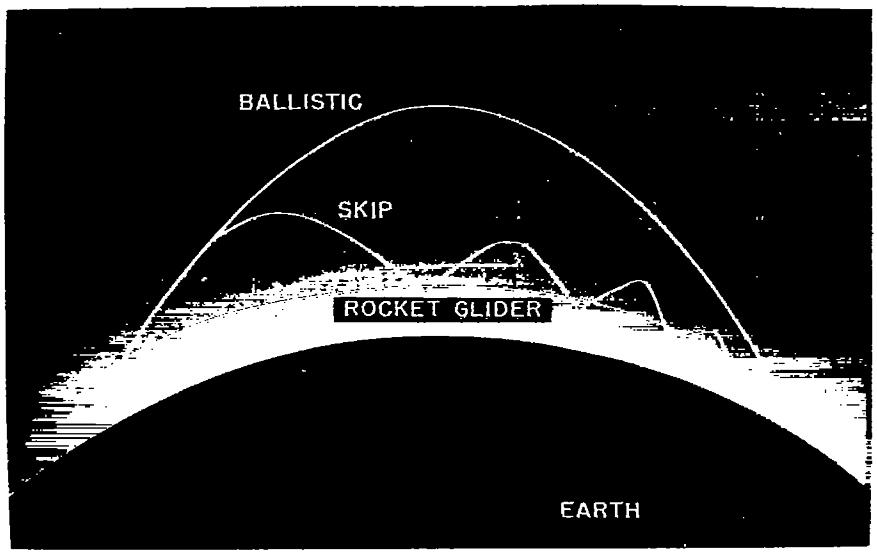

Figure 1

PROPULSIVE RANGE FOR VARIOUS POWER PLANTS

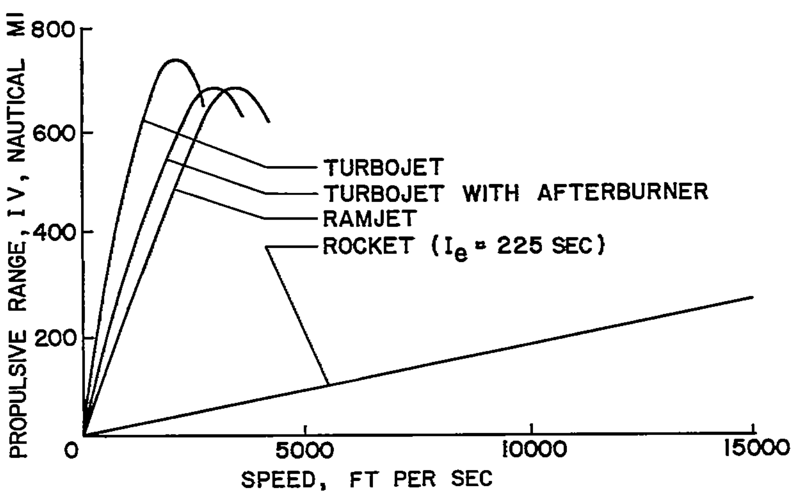

Figure 2

EFFECTIVE LIFT-DRAG RATIO OF HIGH-SPEED VEHICLES: L/D = 2

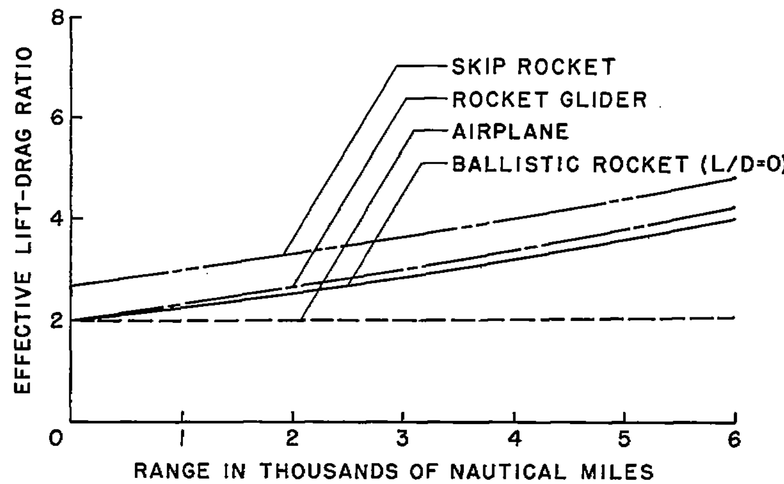

Figure 3(a)

EFFECTIVE LIFT-DRAG RATIOS OF HIGH-SPEED VEHICLES: L/D = 6

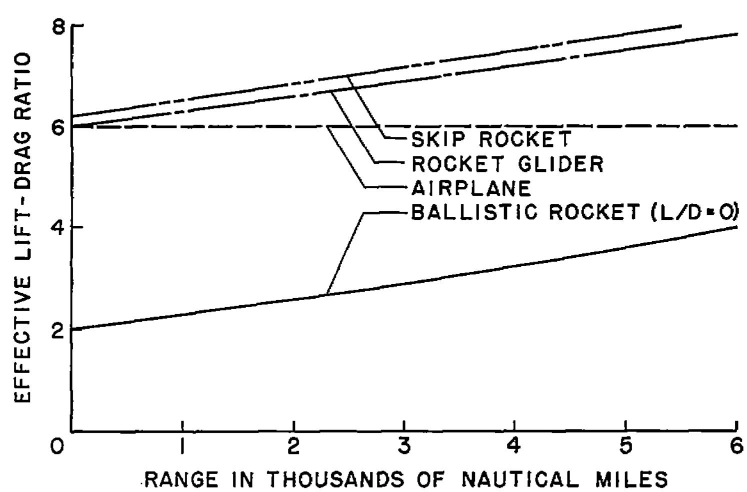

Figure 3(b)

WEIGHT RATIOS FOR HIGH SPEED VEHICLES
L/D = 2

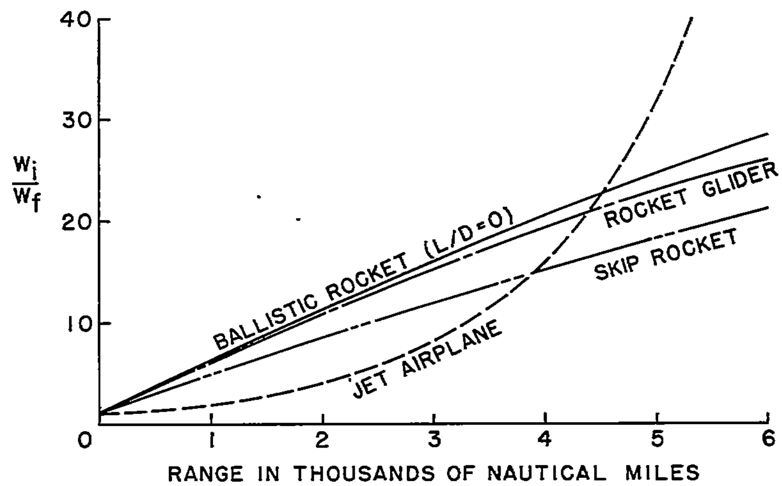

Figure 4(a)

WEIGHT RATIO FOR HIGH SPEED VEHICLES
L/D = 6

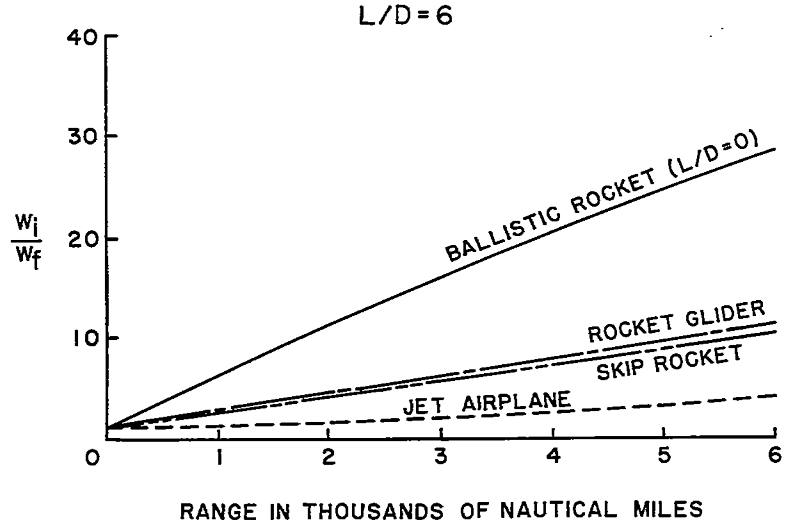

Figure 4(b)

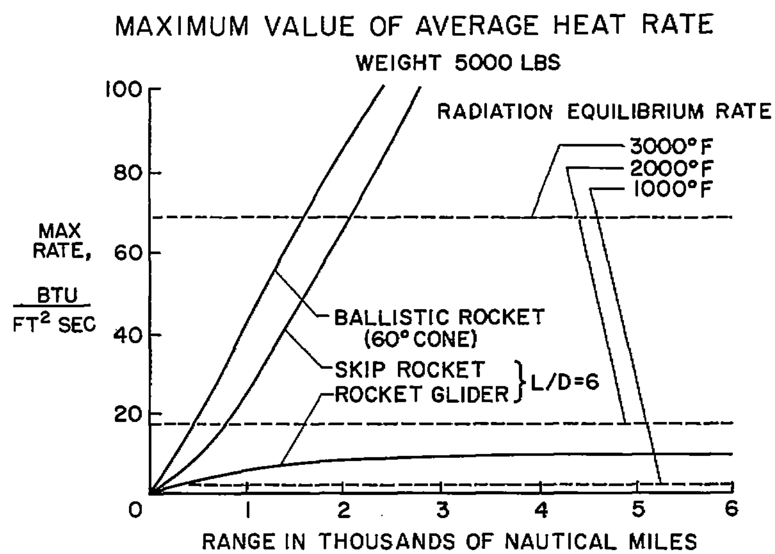

Figure 5

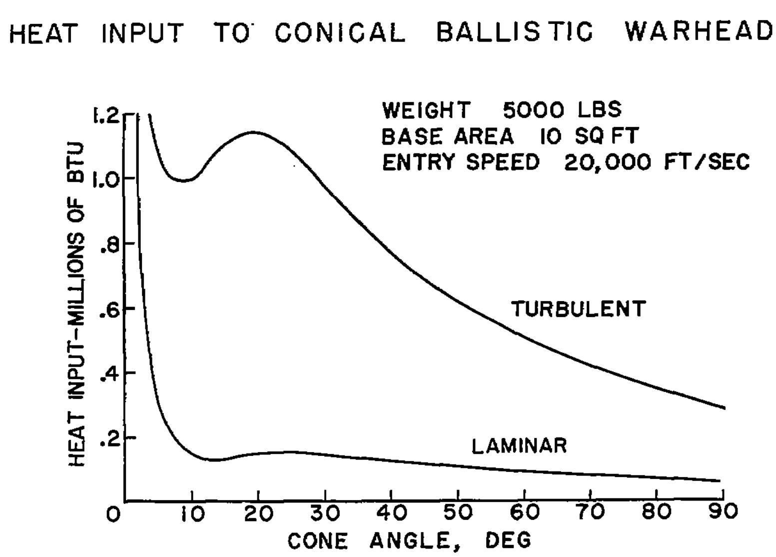

Figure 6

MAXIMUM REYNOLDS NUMBER OF CONICAL BALLISTIC ROCKETS

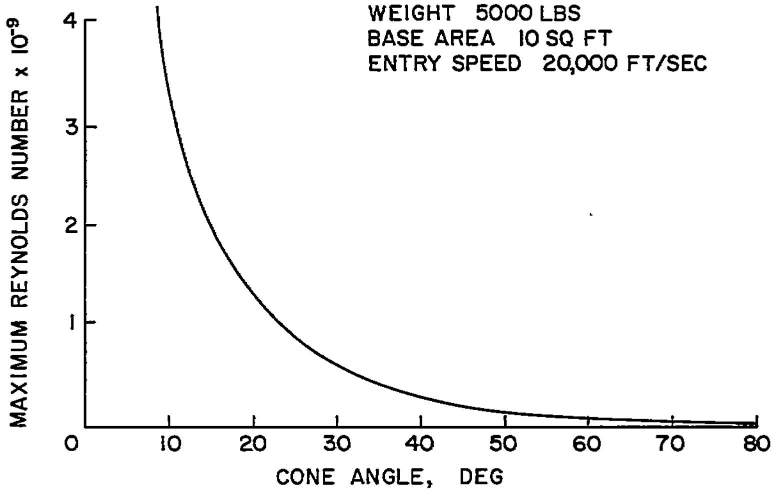

Figure 7

HEAT INPUT PER UNIT FINAL WEIGHT

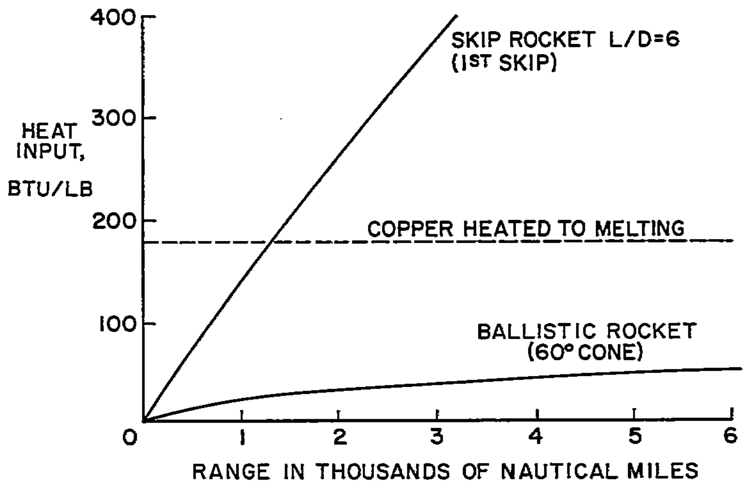

Figure 8

ONE SUGGESTED CONFIGURATION FOR ROCKET GLIDER

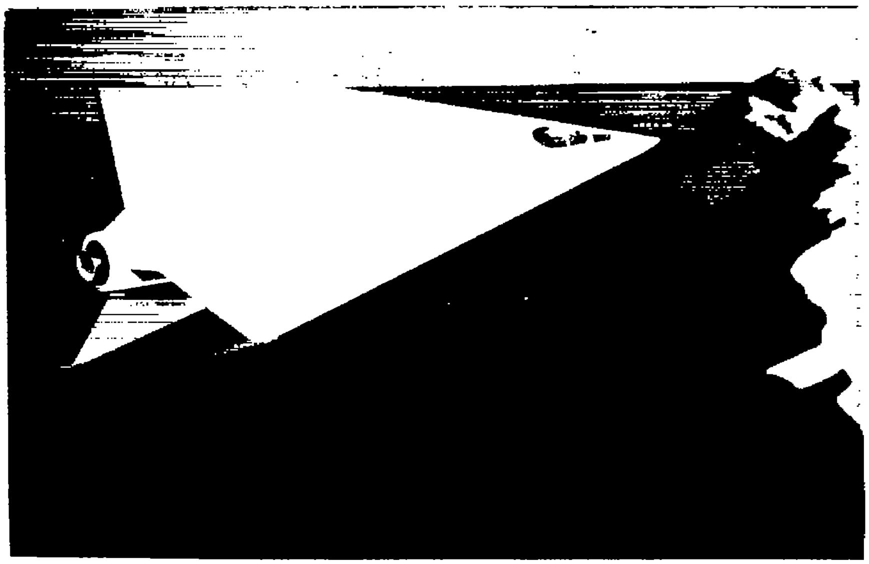

Figure 9

REYNOLDS NUMBER VARIATION WITH SPEED
ROCKET GLIDER (L/D=6, LENGTH=50FT)

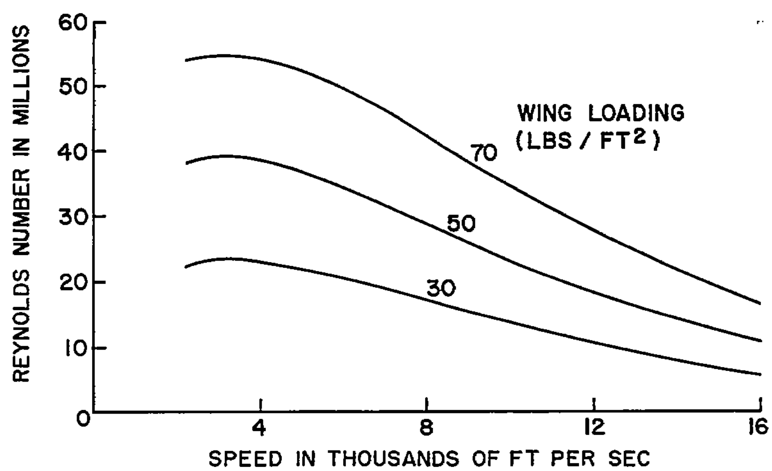

Figure 10

EFFECT OF LEADING-EDGE RADIUS FOR BOOST-SUSTAIN GLIDER

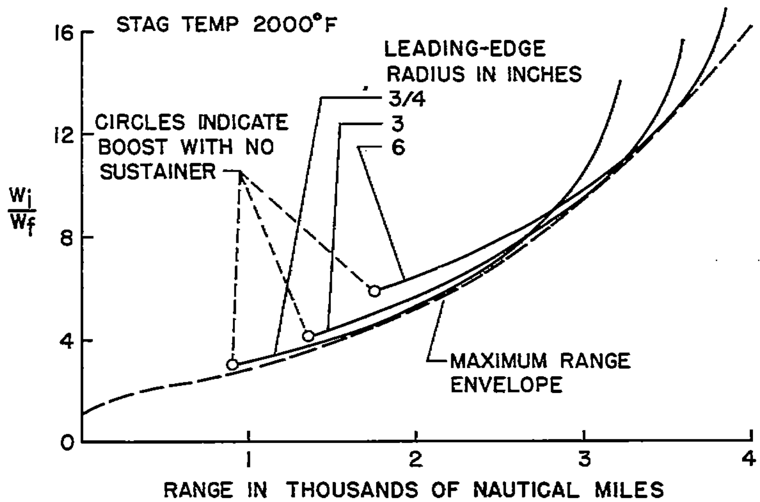

Figure 11

WEIGHT RATIOS FOR RADIATION-COOLED GLIDER

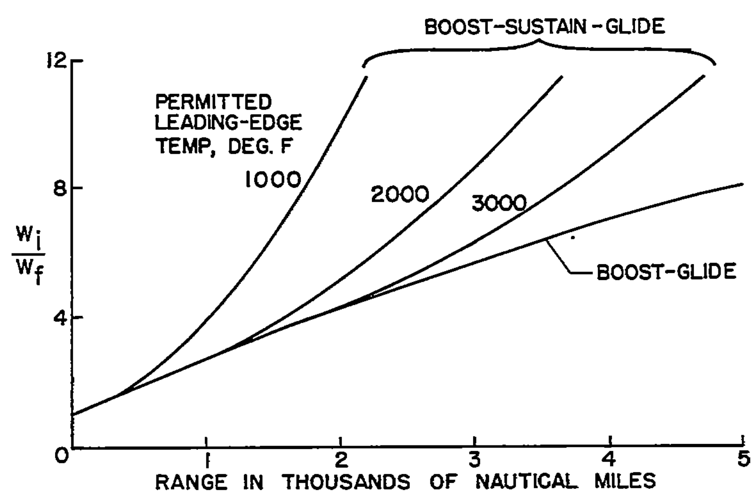

Figure 12

ROCKET GLIDER FOR HIGH LIFT-DRAG RATIO

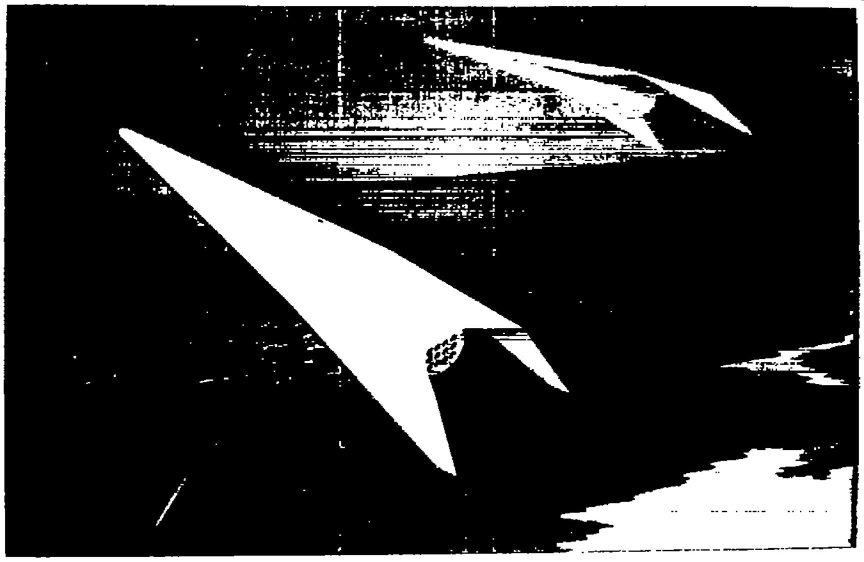

Figure 13

LIFT-DRAG RATIOS FOR A ROCKET GLIDER
AT MACH NUMBER 5

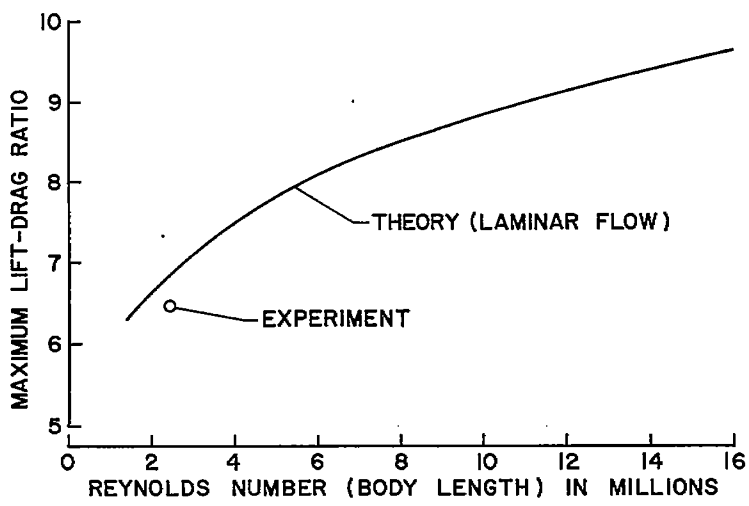

Figure 14

NACA - Langley Field, Va.